

## AUXILIARY ELECTRIC PROPULSION-STATUS AND PROSPECTS.

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### INTRODUCTION

Mission requirements have played a very important role in the direction of development for electric propulsion systems in the millipound and micropound thrust regimes. As early as 1962, it was recognized by many workers in the field that systems in the "micro-thrust" range would be required for a number of spacecraft applications. Probably the first specific case evolved from studies of 24 hour synchronous spacecraft, where reaction jets were found to be necessary for east-west station keeping. As initial missions involved spin stabilized satellites, the problems were complicated by the requirement for pulsing of jets in synchronism with the spin period in order to provide station keeping. Cold gas nitrogen and peroxide systems emerged as the choice for flight application primarily because propellant weight requirements were small and no alternate electric propulsion approach had been developed to a point suitable for flight use.

The early mission studies gave rise to a wide variety of R&D activities to provide electric thrust systems for the future missions (ref. 1). The resistojets, ion microthrusters, and plasma devices were but a few concepts that grew out of the anticipated need for small station-keeping systems. As more serious concern was given to north-south station keeping of spacecraft, attention turned to the higher specific impulse concepts such as bombardment, strip-ion, and colloid thrusters, which could operate at thrust levels from 300 to 500 micropounds. In this case propellant weight has become a strong factor in system selection. A number of application

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studies were performed to assess optimum thrust levels, the advantages of thrust vector deflection, and propulsion system configuration (refs. 2-4). The results of such investigations generally pointed to the strong influence of onboard electrical power for system operation. For example, approximately 75 watts is required to operate a 350 micro-pound cesium ion bombardment microthruster for north-south station keeping of a 24 hour spacecraft. This power drain would be relatively constant for a spacecraft weight of 1,500 pounds.

The serious consideration of gravity-gradient oriented spacecraft in a 24 hour orbit led to yet another mission application for electric propulsion. It was discovered that thrust levels in the range from 10 to 30 micropounds would be required for east-west station keeping in the presence of gravity stabilizing torques. Elastic body effects associated with the long boom extensions on such spacecraft would cause a motion of the center of mass with resulting attitude disturbance torques on the main body during translational corrections due to thruster misalignments. The maximum permissible thrust level for east-west station keeping was established as that value which could tumble the spacecraft within a prescribed time interval. As an outgrowth of this problem, beam-vector thrusters appeared quite attractive for minimizing such attitude disturbances.

The most recent mission applications pertain to attitude fine pointing and slewing of large dish antennas in a 24 hour orbit. The use of electric propulsion devices for such purposes has been extensively studied (refs. 5,6) to establish relative advantages with respect to other competitors such as momentum wheels, control moment gyros, and similar concepts. Such studies have indicated that near-term applications will be decided primarily on inherent reliability of competing approaches rather than size, weight, or power alone. Devices such as momentum wheels have a fixed weight penalty, but require electrical power in amounts dictated by slewing rates, solar pressure profile, and required maneuvering action. Generally speaking, such devices also require onboard reaction jets to enable unloading of wheels intermittently, so that this additional weight penalty should be included in comparisons.

From a mission planning standpoint the future prospects for spacecraft auxiliary electric propulsion appear to be promising. In addition to the follow-on flights into 24 hour orbit involving station keeping, it is foreseen that larger spacecraft in spinning modes will be placed in 24 hour orbit, where precise attitude control will be needed for precession of the spin axis. Three-axis stabilized spacecraft should

become larger in size and weight, necessitating greater total impulse for orbit correction. The MORL is but one example of this trend.

As the observer looks further into the potentialities and future prospects of electric propulsion, the trend appears to be turning strongly in the direction of total systems design approach rather than simple physical tradeoffs. The development costs will definitely play an important role in selection of future concepts. To this point, mission requirements have been primarily limited to applications in a 24 hour orbit. There are at least two other interesting missions that will be mentioned here.

The requirements for Earth observation spacecraft using solar power present unique requirements for orbit control. A typical example is a spacecraft located in a nominal 480 nautical mile circular orbit which is inclined 101.9 degrees with respect to the equator. This orbit provides both sun synchronism to maximize power and a near-synchronous earth ground track at the equator crossings. The mission objective is to maintain the ground track with regard to overlap in picture coverage at successive periodic crossings. The sensitivity to orbit variations is demonstrated by the fact that a 0.6 nautical mile change in semi-major axis produces a 10 nautical mile change in overlap out of a total picture width of 96 nautical miles. Studies of the long term maintenance of such an orbit have shown that onboard reaction jets will be required for mission lifetimes in excess of one year assuming perfect initial injection (ref. 7). The total impulse requirements appear to be well suited to a number of electric propulsion concepts.

Another mission possibility is the Hummingbird satellite, which would be located at one of the libration points in the earth-moon system to serve as a communications relay point in Apollo or post-Apollo missions. Early studies have indicated that electric propulsion may be the preferred approach for orbit maintenance against gravitational perturbations. Further study will be required to obtain a firm direction for development activities.

#### CURRENT STATUS SUMMARY

A brief summary is given here of major directions of hardware development and applications in spacecraft auxiliary electric propulsion. A more detailed description of electric propulsion for prime onboard equipment, and for flight experiments, will be found in ref. 1. The status of prime-onboard operational systems (OS) and flight-experiment systems (FE) is summarized in Table I.

## Operational Flight Systems

The resistojet, in a general sense, represents the most basic attempt to improve propulsion system performance over cold-gas and available chemical devices. Two basic avenues were followed in R&D programs for the resistojet. The first approach treated the thruster as a thermal storage device which would be operated under quasi steady state conditions in a pulse mode manner. This technique maximizes realizable specific impulse for brief pulse durations (of the order of minutes) by the use of stored heat to maintain plenum gas temperature. For long term pulses (continuous operation) such a thruster would degrade in specific impulse to match the end-point energy balance.

Figure 1 shows a single-nozzle version of the TRW Systems resistojet, successfully flown on a Vela III spacecraft in 1965. The design approach was simply to heat the conventional nitrogen propellant to a higher temperature using an electrical heater and thermal insulation on the thruster body. A later version of the TRW resistojet is shown in Figure 2. This represents a multijet thruster flown successfully on an Advanced Vela spacecraft in 1967. Again nitrogen was employed as propellant in a conventional system. Figure 3 presents a single jet version of a thermal storage resistojet built by General Electric and flown on an NRL spacecraft in 1967. This thruster operates from ammonia propellant, which is stored as a liquid in a tank and fed by capillary action to the thruster.

The second design approach for the resistojet was employed by AVCO Corporation as shown in Figure 4. Here, emphasis was placed upon obtaining fast heatup of the propellant contact surface by minimizing the thermal mass of the heating region. The resulting thruster has a general appearance of a hypodermic needle where propellant is injected through the needle and the ends of the needle are resistively heated. Experimental systems of this type were built and flown on the ATS B and C spacecraft, and advanced versions will be prime-onboard equipment on the ATS D and E spacecraft.

The tradeoff factors between the two design approaches depend heavily upon the intended use, specifically the pulse length and required duty cycle. For attitude control having very low duty cycle, the pulsed-mode resistojet offers significant power savings. For long-term pulsing or high duty cycle the thermal-storage resistojet appears attractive because of the higher obtainable specific impulse.

## Flight Experiment Systems

In addition to the resistojet programs which have attained a flight status, there are a number of flight experiment systems presently under development to assess their performance capabilities in space for mission functions similar to those previously described. These systems are scheduled to fly in the Air Force multi-purpose synchronous satellite program (AFMS), on the ATS D and E spacecraft, and in the LES program (ref. 8).

The General Electric Company ammonia thermal-storage resistojet shown in Figure 5 is designed to demonstrate multiple (redundant) exhaust jets for station-keeping functions in the AFMS program. GE is also developing for NASA Goddard a similar resistojet where the nozzle configuration will provide combined attitude control and station-keeping from a single thruster module and heater. Also included in the AFMS program are the Electro-Optical Systems, Inc. cesium electron-bombardment thruster, for north-south station keeping, shown in Figure 6; and the TRW Systems liquid-spray charged-particle thruster for east-west station keeping and yaw control, shown in Figure 7.

Figure 8 shows a cesium contact ion microthruster with beam vectoring capability which is now under development at Electro-Optical Systems for testing on the ATS D and E spacecraft. In addition to evaluating system behavior this experiment will also permit a demonstration of east-west station keeping thrust levels on a gravity gradient stabilized spacecraft.

Pulsed-plasma thrusters have also reached a development stage where flight experiments can be scheduled. The Republic Aviation Division of the Fairchild Hiller Corporation is preparing a solid-propellant pulsed-plasma thruster system for the LES flight program.

The ATS F and G spacecraft programs presently under study might be vehicles for testing advanced versions of a number of thruster systems discussed so far.

## ADVANCED CONCEPTS

There are a number of advanced concepts in auxiliary electric propulsion that have promise of providing superior performance, greater reliability, and better matching with the spacecraft interface. Some of these concepts have been mentioned previously in reference 9, but are discussed further here, and in addition, some new concepts are presented.

## Radioisojet

A direct alternate to the resistojet class of thrusters is the Radioisojet, which employs a radioisotope thermal source in place of the electrical heater (ref. 10). Early development work for such thrusters has been carried out by General Electric Company and TRW Systems. The GE approach employs promethium-147 oxide as the heat source, and the thruster configuration is shown in Figure 9. The TRW Systems design approach employs Pu-238 as the heat source in a thruster configuration somewhat similar to that previously shown. The major factors in selection of best radioisotope lie in requirements for radiation shielding and aerospace safety for a flight unit.

During tests of the Radioisojet, ammonia propellant temperatures of  $1270^{\circ}\text{K}$  were attained. For reasons that will become clear in the ensuing text, the radioisojet technology is an important and essential element in a number of advanced electric propulsion concepts.

## MORL Resistojet

The manned orbiting research laboratory (MORL) is a possible unique application for electric propulsion in the foreseeable future. Studies by the McDonnell Douglas Corporation (ref. 11) have shown that a resistojet propulsion system (Figure 10) operating with biowaste propellant would serve well for drag cancellation and for attitude control in conjunction with control-moment gyros. The EC/LS system for a 6-crew MORL could consist basically of an electrolysis unit to produce oxygen from water supplied periodically to the spacecraft. A major constituent of the biowaste would be carbon dioxide gas, which would provide more than sufficient propellant for all propulsion requirements.

Because of oxidizing impurities in the carbon dioxide biowaste, it appears that resistojet operation may be limited to  $3000^{\circ}\text{R}$  plenum temperature. With 40 psia plenum temperature, the performance of such a resistojet would be about 4 watt(electric)/millipound at a specific impulse of 178 seconds (ref. 11).

A possible way of improving this performance is by combining Radioisojet technology with the carbon-dioxide resistojet in the ATEP concept (augmented thermally electric propulsion). Principles and advantages of the ATEP concept are described and analyzed in ref. 12. The diagram in Figure 11 illustrates a hypothetical application to the MORL electric propulsion system. If the radioisotope heater performance could approach that of the Radioisojet, then the

carbon dioxide could be thermally heated to 1200°K, then electrically heated to the 3000°R compatibility limit. As shown in Figure 12, the ATEP concept with radioisotope pre-heating of the propellant might lower the power/thrust to 1.7 watt(electric)/millipound. This reduction in electric power could be very significant in missions of the MORL type.

### Supersonic Heat Addition in Electrothermal Thrusters

Another application of the ATEP concept is heat addition to supersonic streams by electrical means. In principle, significant increases in specific impulse are possible, without incurring dissociation or ionization losses in the near frozen flow to be expected in electrothermal thrusters. Analyses reported in refs. 13 and 14 have shown that the mode of heat addition has a very great effect on the nozzle area ratio that is required to reach significantly higher exhaust velocities. The best heating mode reported to date is a linear relation between heat addition and nozzle cross-sectional area, and this mode is used here merely for want of better information.

Theoretical performance of a lithium electrothermal thruster with supersonic heat addition is shown in Figure 13. An area ratio of 100 is assumed for the nozzle, as being representative of an optimum realistic design (ref. 15). Plenum temperature is assumed to be 2500°K, which is in keeping with demonstrated resistojet temperatures (ref. 16), and a negligible exhaust pressure is assumed. Even with the non-optimum mode of supersonic heat addition, significant increases in specific impulse are indicated as the total temperature is increased. This increase in specific impulse appears possible with negligible change in power/thrust. With a fully optimized mode of heat addition, even better performance is anticipated, as discussed in ref. 14.

Even more dramatic improvement in performance is predicted when a combination of thermal augmentation and supersonic heat addition is employed, as shown in Figure 14. Much of this improvement is due to the high heat of vaporization of lithium. As with the carbon-dioxide resistojet, radioisotope thermal heating of the propellant to 1200°K is assumed, which is within the present state-of-the-art for Radioisotjet technology. Further improvement in Radioisotjet design could allow increased thermal heating of the lithium propellant, with attendant reduction of the electric power requirements. In addition, further improvement in performance with supersonic heat addition should be possible with optimized modes of heat addition.

## Higher-Voltage Solar-Cell Arrays

Advantages of higher-voltage solar-cell arrays are discussed in ref. 9 with regard to system weight and reliability of ion thrusters. As more flight-system experience is gained, it becomes increasingly evident that the power conditioning needed to provide kilovolts of electric power for electrostatic thrusters is a serious detriment to system reliability. Operation of solar-cell arrays at hundreds of volts could completely eliminate power conditioning. Some work is presently being done to investigate the feasibility of operating the solar arrays at higher voltage (ref. 17).

In the specific impulse range normally dominated by resistojets, it appears possible to obtain good performance from the liquid-spray charged-particle thruster with voltages in the hundreds of volts. As shown in Figure 15, a solar-cell array voltage of 400 volts could provide a specific impulse of 280 seconds with charged-particle  $q/m = 10,000$  coulomb/kg, which is a presently attainable value (ref. 18). Performance of the liquid-spray thruster at 280 seconds specific impulse is definitely competitive with conventional resistojets. If even higher solar-cell-array voltages were possible, the liquid-spray thruster could exceed the specific impulse of present-day resistojets in the low micro-pound thrust range.

Electrostatic ion thruster performance may be very attractive with only moderate increases in solar-cell-array voltage, as shown in Figure 16. The hypothetical thruster used for these theoretical calculations was assumed to have the demonstrated performance of a cesium hollow cathode, with total-throughput ion optics (ref. 19), and adequate power density by virtue of a composite-grid accelerator structure (ref. 20). Satisfactory operation with composite grid accelerators has been achieved in 100 hour tests with thrusters from 5 cm to 30 cm in size, and with net accelerating voltages from 400 to 1000 volts (ref. 21). Power/thrust for the hypothetical thruster with other alkali propellants were calculated with the assumption that the cathode power consumption (ev/ion) would be proportional to the first ionization potential of the propellant.

From this brief examination of the performance of two extremes of electrostatic thrusters (with respect to charge/mass of propellant), it is evident that much would be gained if solar-cell arrays could be operated at hundreds of volts.



## The ACCENT System

A new concept in auxiliary electric propulsion is called ACCENT (for autogenically-controlled-cesium-, or colloid-, electro-nuclear-thrust system). This concept offers promise of significant reductions in propellant mass and in demands on the spacecraft power system, and by virtue of the small number of components, the ACCENT system offers greatly improved reliability (ref. 22). High-voltage dc electric power is generated in a radioisotope electrogenerator (REG) as described in refs. 22 and 23, thereby eliminating the complex power conditioning needed for conventional electrostatic ion and colloid thrusters, and for pulsed-plasma thrusters.

In ACCENT systems having contact-ion thrusters, radioisotope heating of the ionizer would be required. Low-voltage power for the neutralizer and control circuitry could be provided by incorporating radioisotope thermoelectric generators (RTG).

The REG performance improves as operating voltage is increased, reaching a maximum efficiency at about 50,000 volts when promethium-147 is the fuel. This high-voltage capability makes the ACCENT system especially well suited to the liquid-spray charged-particle thruster. With the bi-polar liquid-spray thruster, there is no neutralizer, and propellant-feed power requirements could be supplied by waste heat from the REG. Low-voltage power for control circuitry could be drawn from the spacecraft power system, or could be generated by an integral RTG within the REG containment vessel. Hypothetical ACCENT systems with 20 micropound and 350 micropound bi-polar liquid-spray thrusters are shown in Figures 17 and 18. Although REG of such designs have not been developed, there is sufficient experimental background to indicate fundamental feasibility (refs. 22 and 23). In fact, some characteristics of the REG would be especially compatible with electrostatic thrusters; for instance, an electric breakdown in the thruster would merely drain the REG much like a capacitor discharge, thereby automatically limiting breakdown damage to the thruster.

Because of the capacitor-like action of the REG, the ACCENT system should be well suited to pulsed-plasma thrusters. There are a number of pulsed-plasma thrusters under development at present. These can be characterized in scope by the solid-propellant Fairchild Hiller thruster described previously, and by the gas-propellant inductive-accelerator system under development at TRW Systems (ref. 24). These thrusters require a capacitive discharge for each thrust pulse, and this discharge could be provided by the REG in an ACCENT system.

With the virtual elimination of all power demand on the spacecraft system, and with reliability through simplicity, the ACCENT system appears to be an advanced concept of great interest.

#### FUTURE TRENDS

The usual criteria for selecting most promising design concepts is basically one of comparing physical parameters such as size, weight, power, and development status. Experience with flight systems and those under development for flight experiments has shown that use of such physical criteria must be replaced with a total systems design evaluation which includes the categories of reliability, design complexity, and cost. For example, the existing resistojet systems rely on an electrical heater for improvement in specific impulse. This heater must ultimately demonstrate high-temperature, long-life capability in order to maintain a competitive position among other approaches. In contrast, Radioisotjet uses a radioisotope heat source of inherent high reliability. However, now the problems shift into radiation shielding, spacecraft interfaces, and aerospace safety. Both of these systems require a propellant on-off solenoid valve, which itself must be treated as one of the most important design factors.

For the ion and colloid thrusters, the problems become high-voltage circuitry, electromagnetic interference, reliability of heater elements and electrode degradation. The pulsed plasma devices which are valveless still must deal with electrode deposition and with high-voltage circuitry to obtain competitive performance. A number of design options have been studied to remedy the major reliability problem areas being faced by the conventional ion and colloid systems. For example, the neutralizer could be eliminated on the colloid thruster by employing a bipolar concept which groups positive and negative needle arrays. Either high-voltage solar-cell arrays, or the ACCENT concept could eliminate most of the complex power conditioning circuitry needed for either engine type. In addition, the ACCENT system offers complete independence from the spacecraft power system by virtue of complete self-containment. There probably will be an upper practical limit to solar-cell array voltage, which will prevent use of this concept for all applications. Similarly, radiation shielding and aerospace safety requirements probably will limit the thrust range of ACCENT systems.

None of the advanced concepts discussed in this paper are offered as universal solutions for auxiliary propulsion applications. Each has particular advantages and regimes of operation, and all are deserving of some further study to assess their final value to auxiliary electric propulsion.

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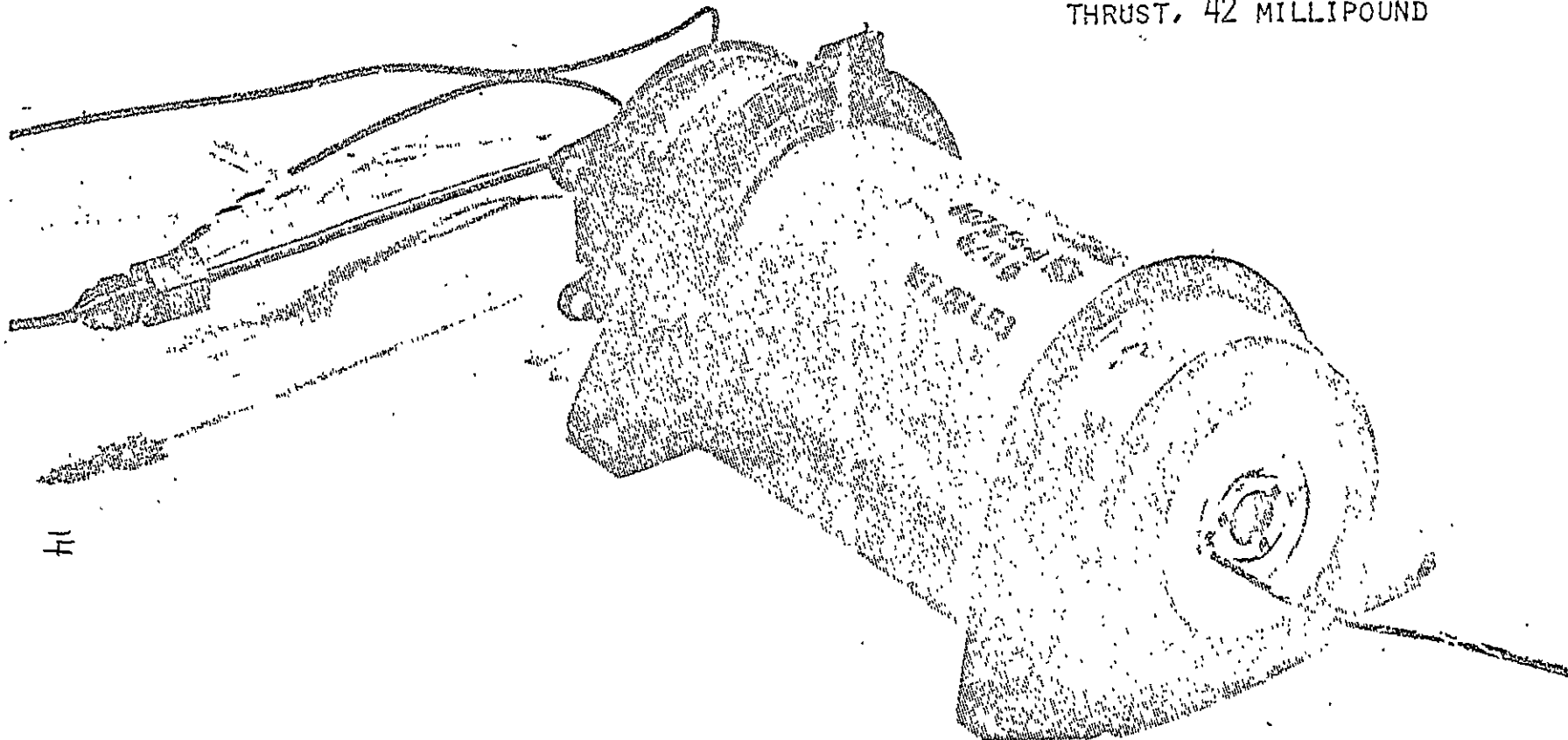
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TABLE I - Summary of prime-onboard operational systems (OS), and flight-experiment systems (FE).

	'65	'66	'67	'68	'69
<u>VELA-III</u>					
TRW RESISTOJET: NITROGEN, SINGLE-JET, THERMAL-STORAGE	OS				
<u>VASP</u>					
TRW RESISTOJET: NITROGEN MULTI-JET, THERMAL-STORAGE				OS	
<u>NRL</u>					
GE RESISTOJET: AMMONIA, SINGLE-JET, THERMAL STORAGE				OS	
<u>ATS B, ATS C</u>					
AVCO RESISTOJET: AMMONIA, SINGLE-JET, PULSED-MODE				FE	
<u>AF MULTI-PURPOSE</u>					
GE RESISTOJET: AMMONIA, MULTI-JET, THERMAL-STORAGE					FE
EOS ELECTROSTATIC THRUSTER: CESIUM, ELECTRON-BOMBARDMENT					FE
TRW ELESTROSTATIC THRUSTER: GLYCEROL, LIQUID-SPRAY					FE
<u>ATS D, ATS E</u>					
AVCO RESISTOJET: AMMONIA, SINGLE-JET, PULSED-MODE					OS
EOS ELECTROSTATIC THRUSTER: CESIUM, CONTACT-ION, TWO-AXIS THRUST VECTORING					FE
<u>LES</u>					
REPUBLIC AVIATION PLASMA THRUSTER: SOLID PROPELLANT, PULSED-MODE					FE

# VELA III RESISTOJET

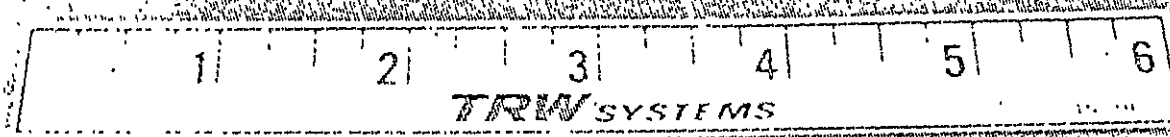
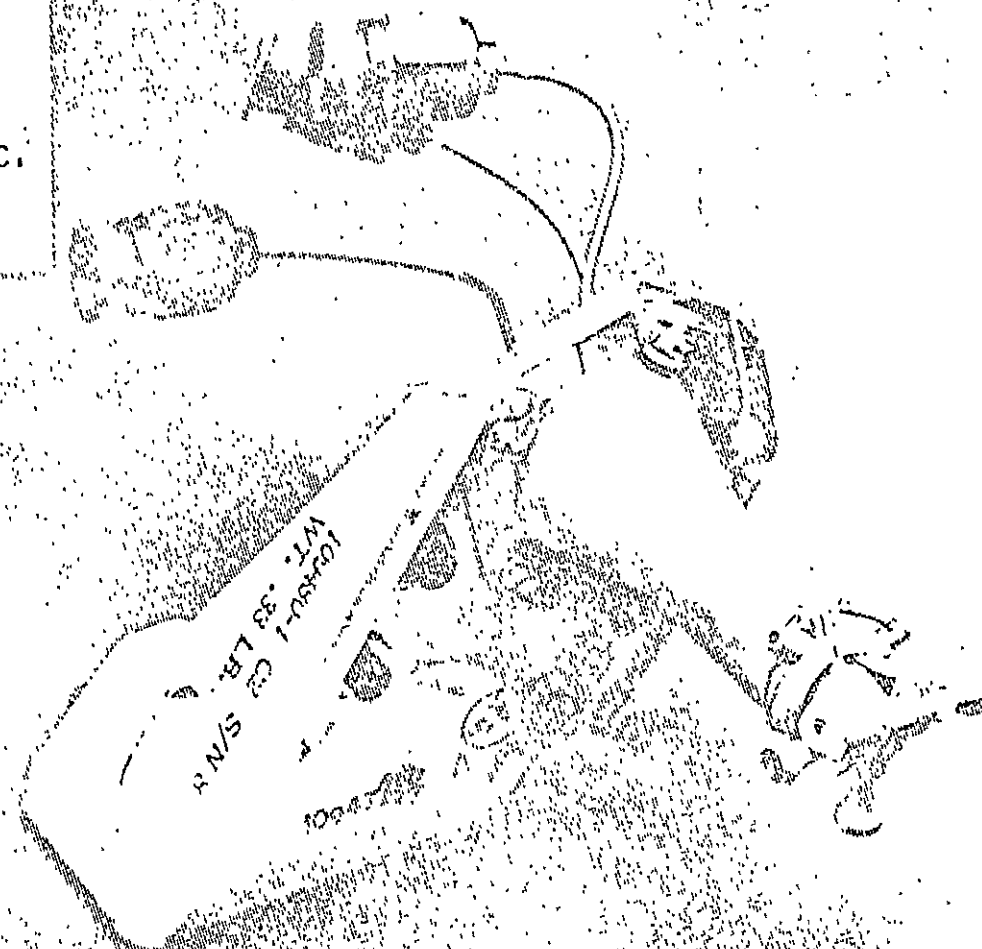
NITROGEN PROPELLANT  
SPECIFIC IMPULSE, 123 SEC  
POWER, 42 WATTS  
THRUST, 42 MILLIPOUND



1 2 3 4 5  
**TRW SPACE TECHNOLOGY LABORATORIES**

FIG 1 - TRW RESISTOJET FOR VELA III SPACECRAFT, SINGLE NOZZLE.

MULTI-NOZZLE RESISTOJET  
FOR ADVANCED VELA  
NITROGEN PROPELLANT  
SPECIFIC IMPULSE 132 SEC.  
POWER, 17 WATTS  
THRUST, 20 MILLIPOUNDS



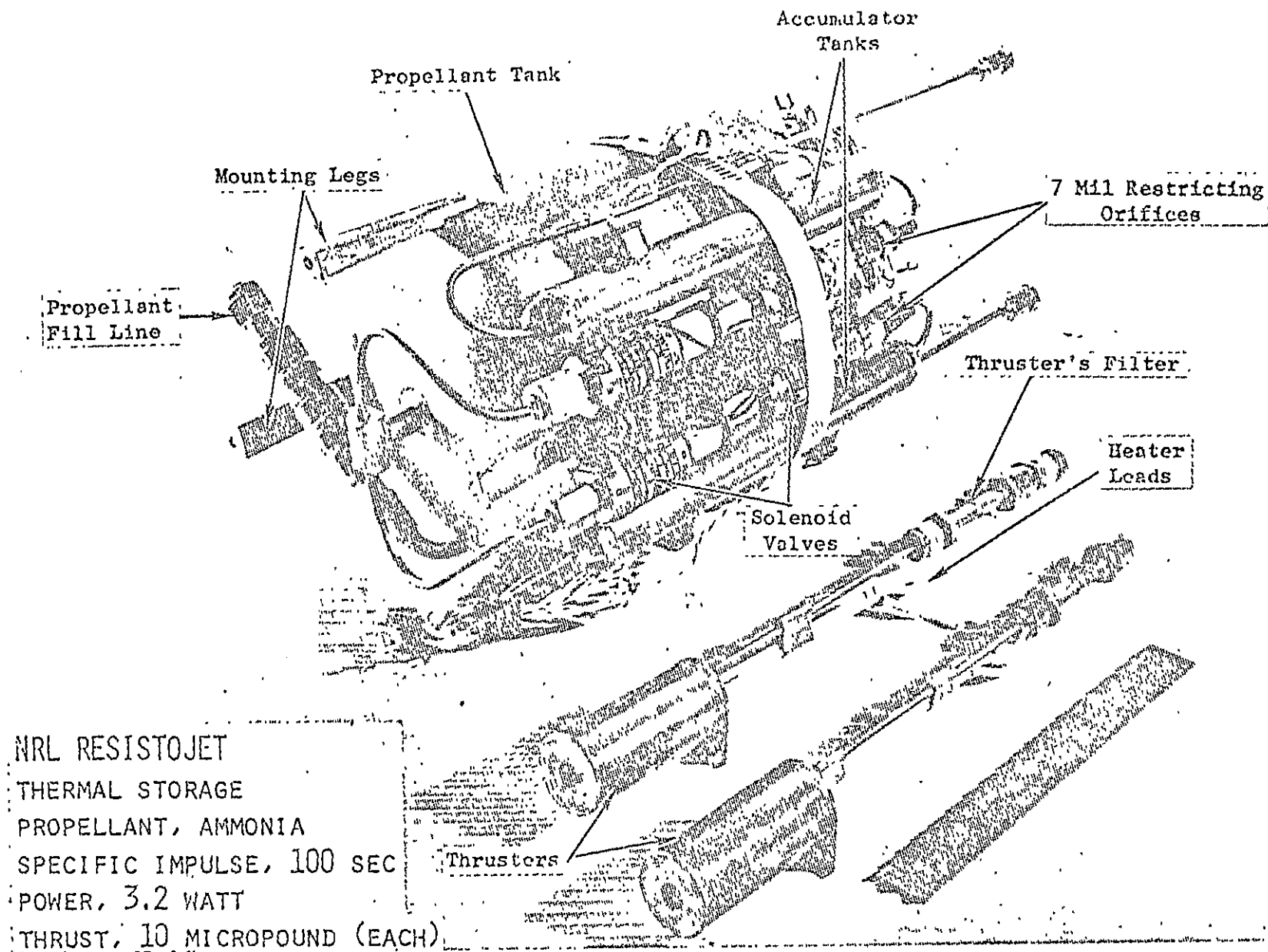


FIG. 3 - GE RESISTOJET FOR NRL SPACECRAFT.



# ATS-C DUAL RESISTOJET

PROPELLANT, AMMONIA

SPECIFIC IMPULSE, 110 AND 140 SEC.

POWER, 9 AND 10 WATTS

THRUST, 10 AND 100 MICROPOUNDS

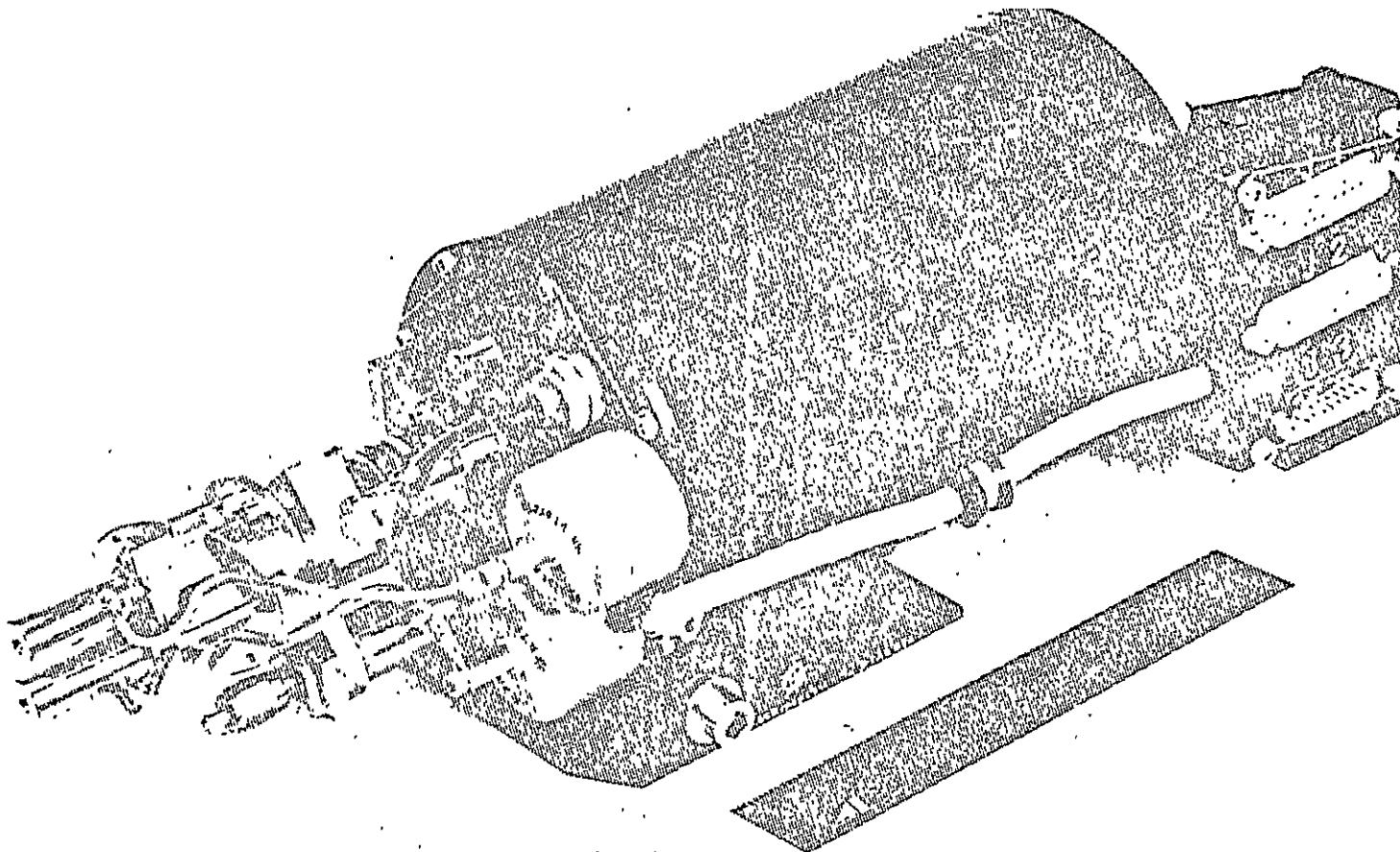


FIG. 4 - AVCO RESISTOJET FOR NASA ATS C SPACECRAFT.

3-NOZZLE THERMAL-STORAGE RESISTOJET  
FOR AIR FORCE MULTI-PURPOSE SATELLITE  
PROPELLANT, AMMONIA  
CATALYST BED IN FLOW PASSAGE

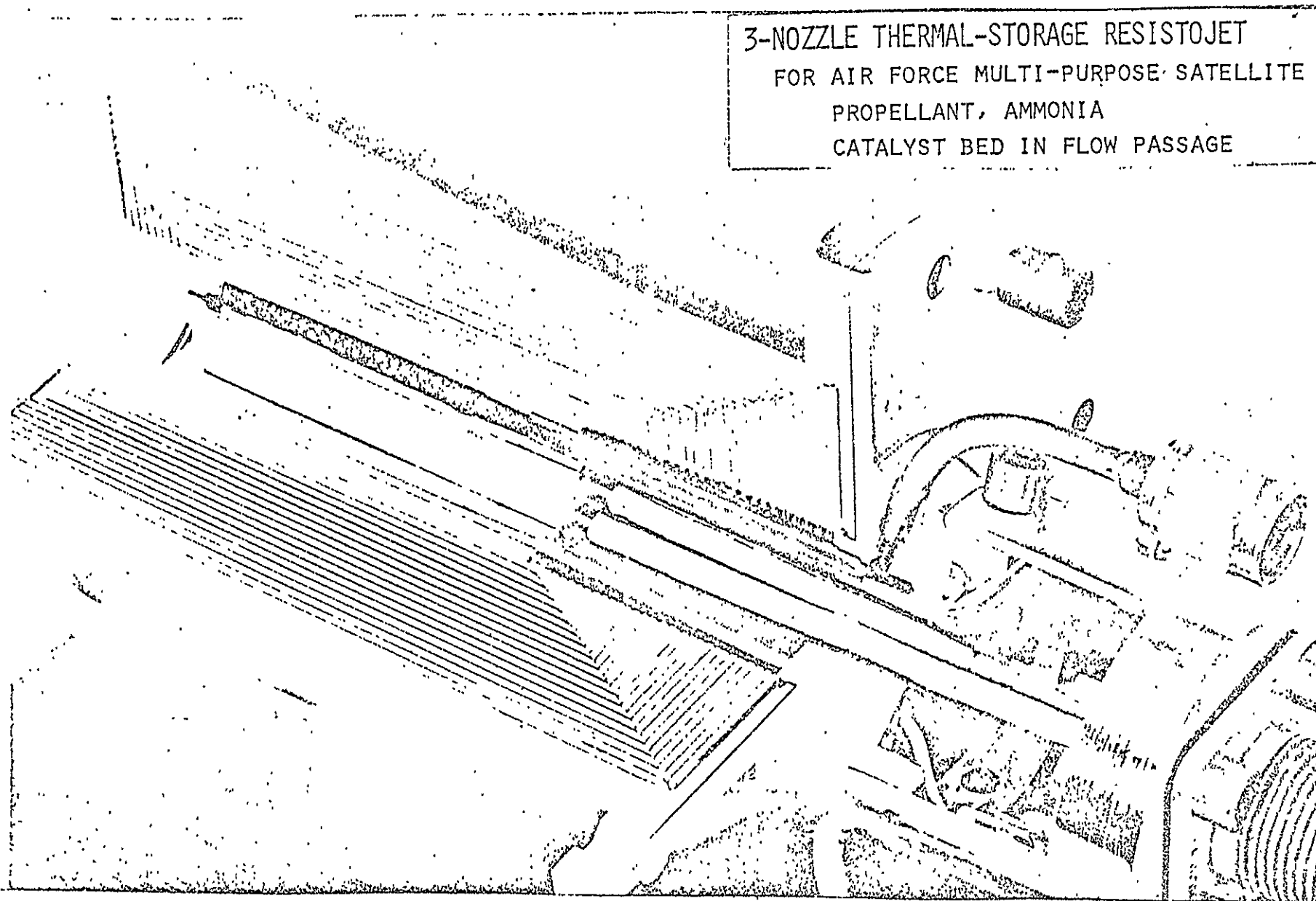


FIG. 5 - GE RESISTOJET FOR AIR FORCE MULTI-PURPOSE SATELLITE PROGRAM. MULTI-NOZZLE;

ELECTRON-BOMBARDMENT THRUSTER  
FOR AIR FORCE MULTI-PURPOSE SATELLITE  
PROPELLANT, CESIUM  
SPECIFIC IMPULSE, 3000-5000 SEC.  
THRUST, 50 MICROPOUND TO 10 MILLIPOUND

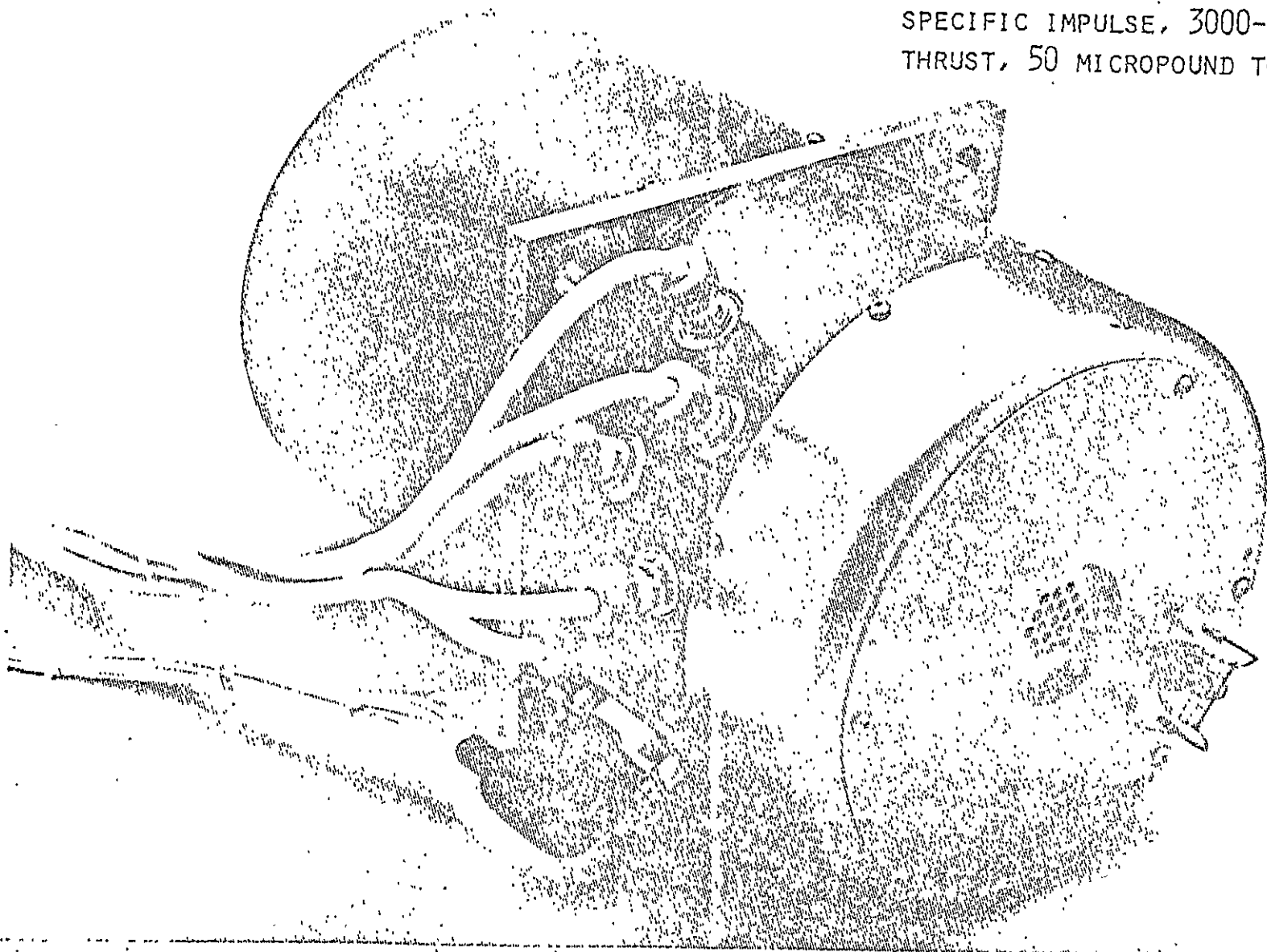


FIG. 6 - EOS CESIUM BOMBARDMENT THRUSTER FOR  
AIR FORCE MULTI-PURPOSE SATELLITE PROGRAM.

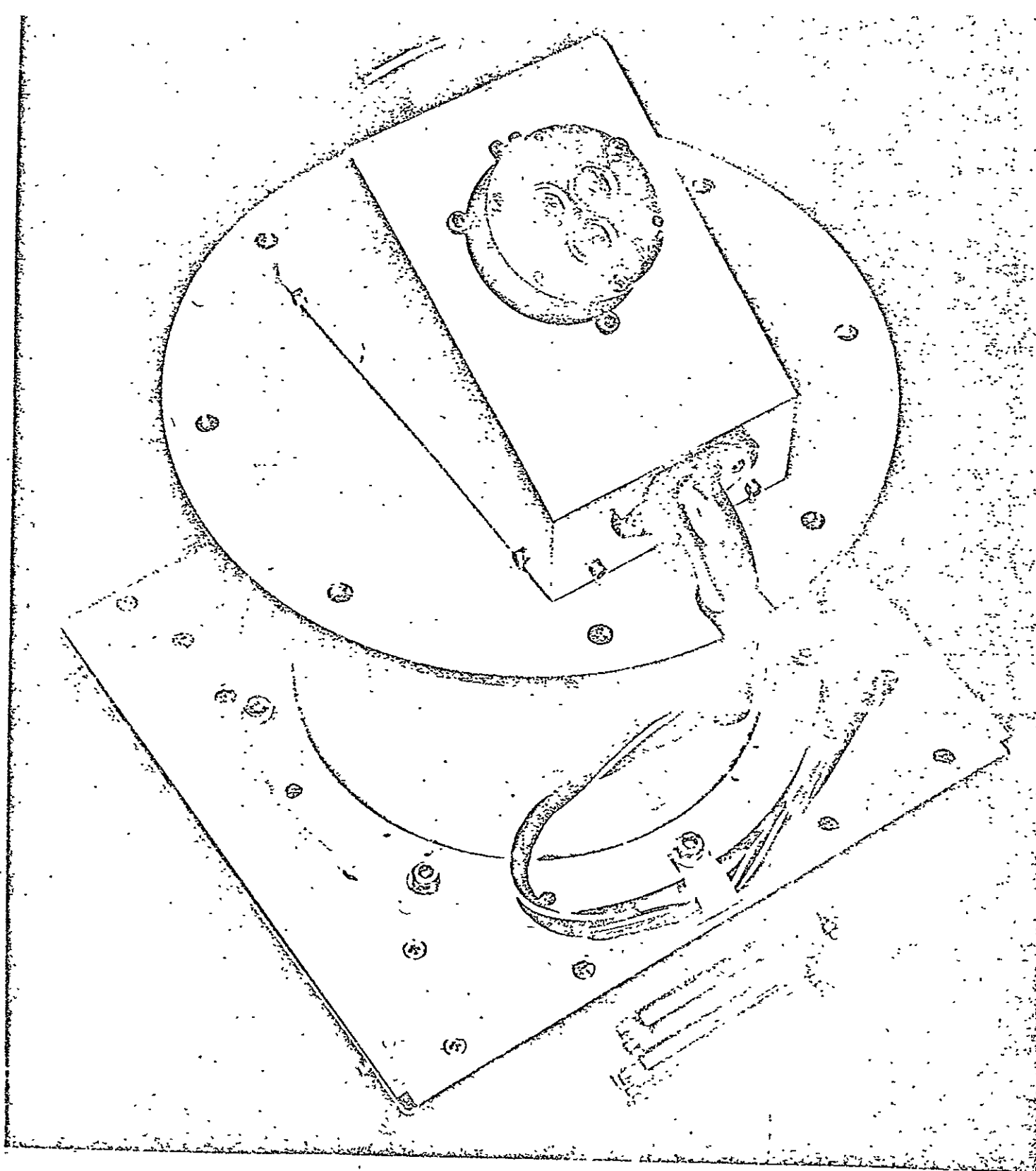
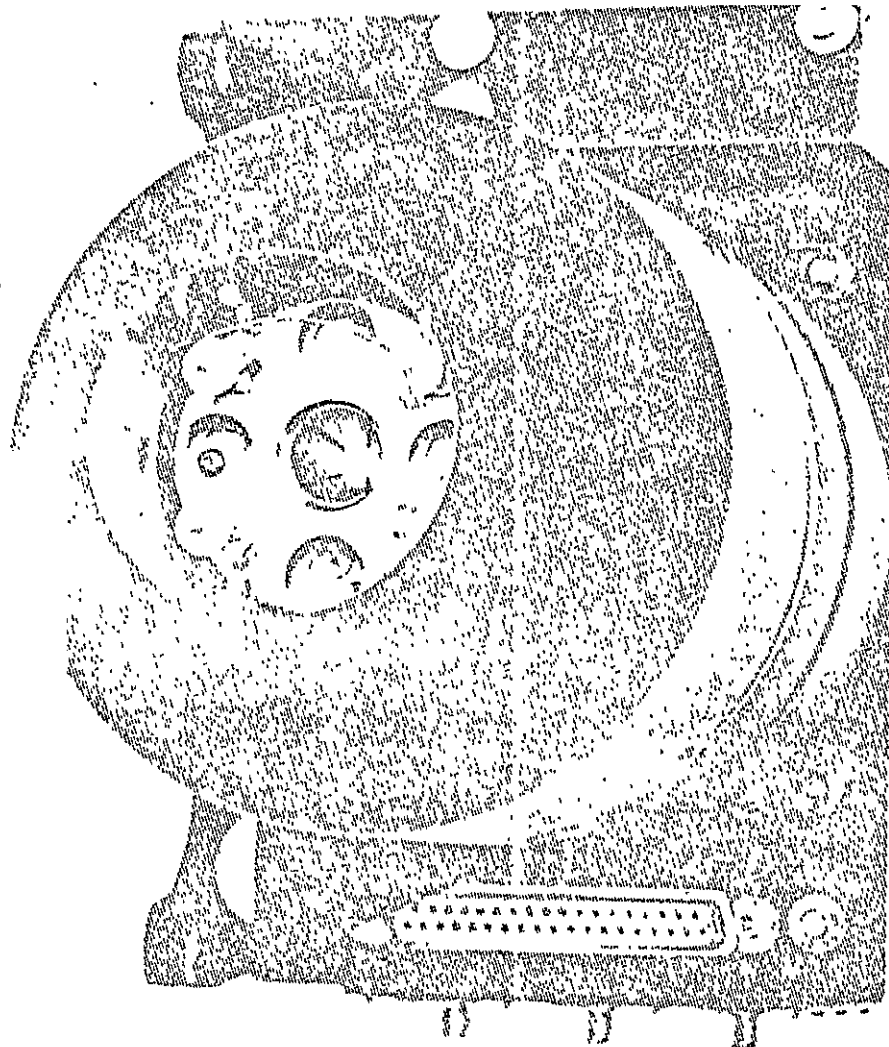
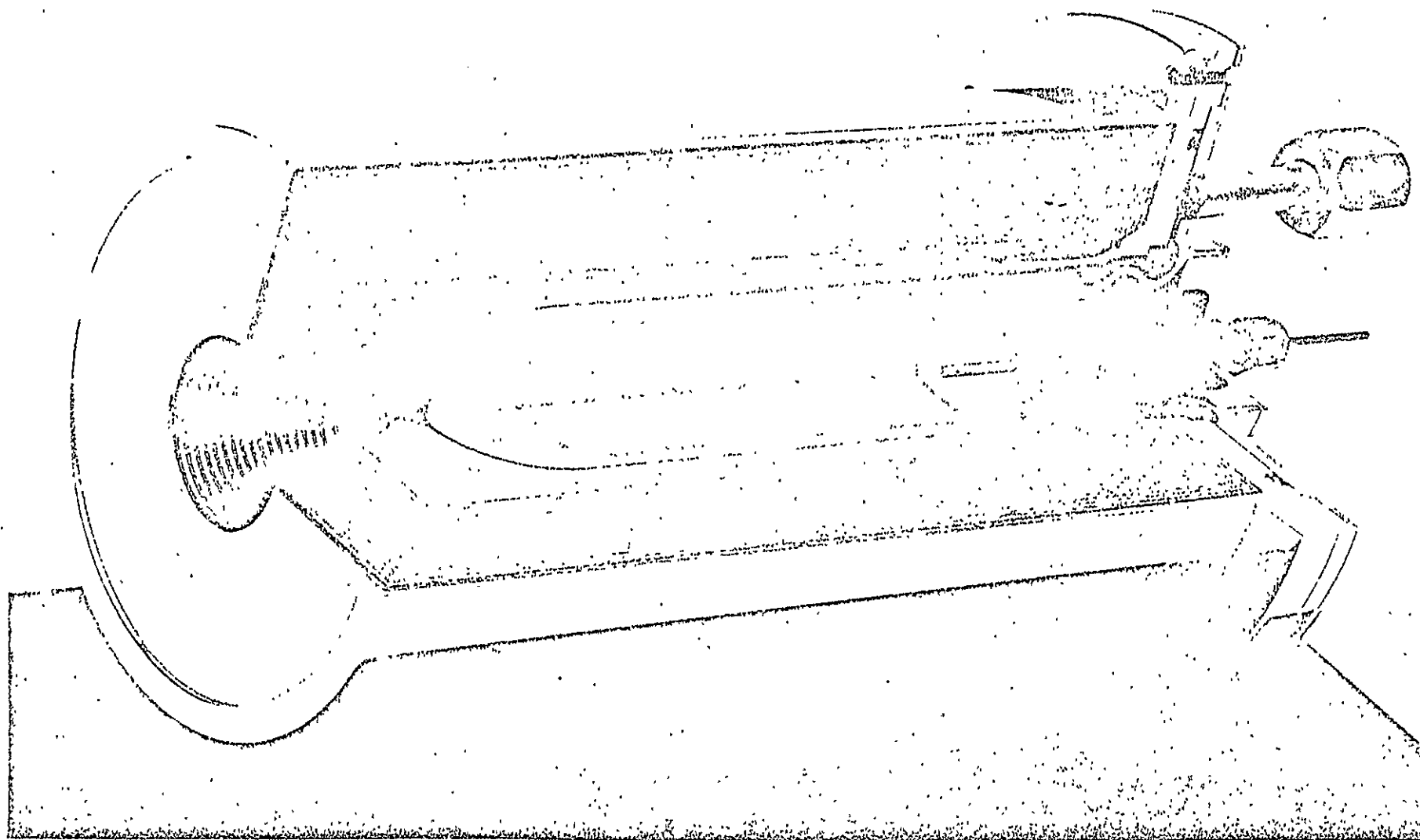


FIG. 7 - TRW SYSTEMS LIQUID-SPRAY CHARGED-PARTICLE THRUSTER FOR AIR FORCE MULTI-PURPOSE SATELLITE PROGRAM. PROPELLANT GLYCEROL; SPECIFIC IMPULSE, 600-1000 SEC.; POWER, 5.2 WATTS; THRUST, 20 MICROPOUNDS.



CONTACT-ION THRUSTER  
FOR ATS-D  
PRECISION THRUST VECTORING  
PROPELLANT, CESIUM  
SPECIFIC IMPULSE, 6700 SEC  
POWER, 35 WATT  
THRUST, 20 MICROPOUND

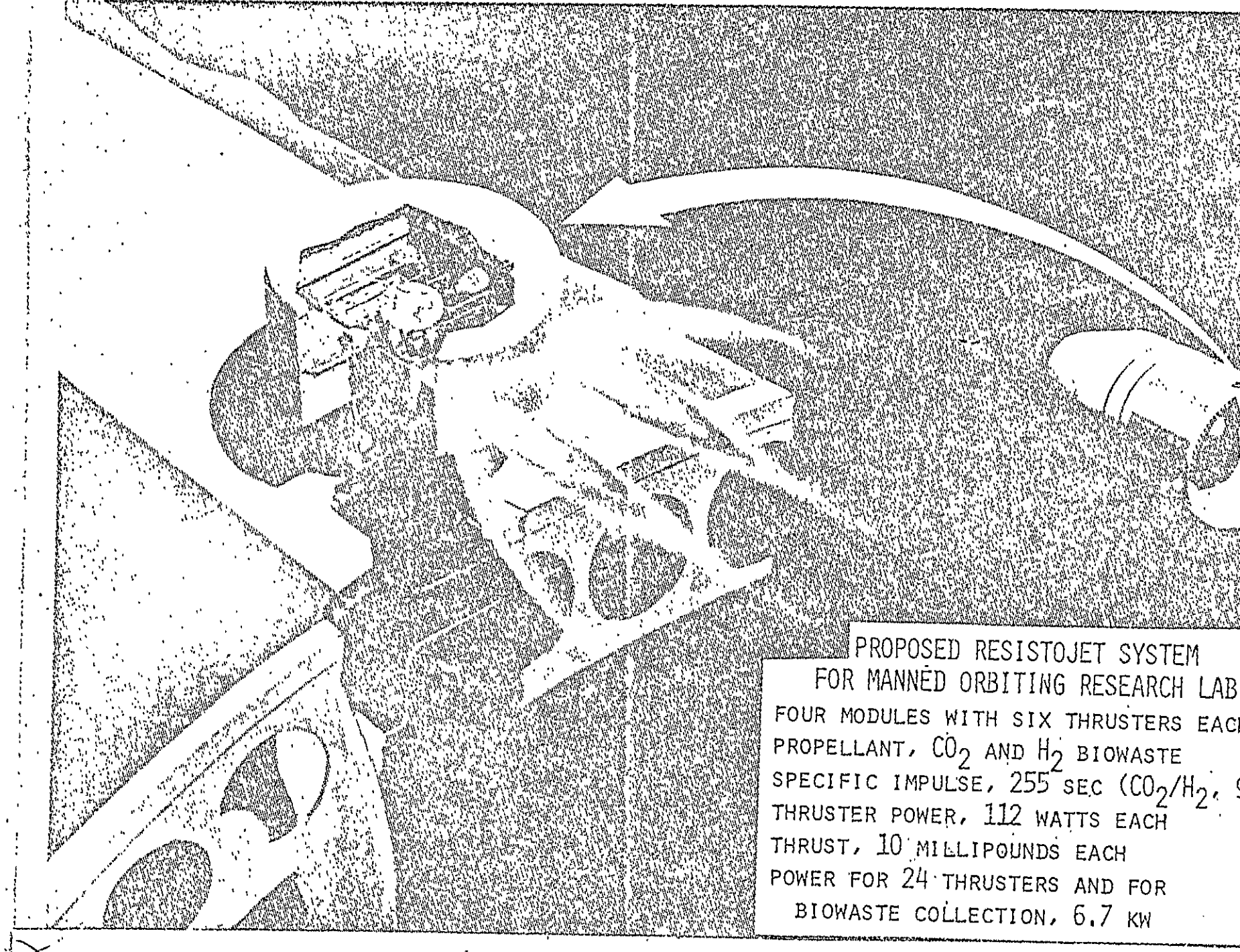
FIG. 8 - EOS CESIUM CONTACT ION THRUSTER FOR NASA ATS D SPACECRAFT.



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FIG. 9 - GE RADIOISOJET DEVELOPED FOR NASA GODDARD.  
RADIOISOTOPE, 60-WATTS PROMETHIUM-147; PROPELLANT,  
AMMONIA; THRUST, 25 MILLIPOUNDS; SPECIFIC IMPULSE,

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PROPOSED RESISTOJET SYSTEM  
FOR MANNED ORBITING RESEARCH LAB  
FOUR MODULES WITH SIX THRUSTERS EACH  
PROPELLANT,  $\text{CO}_2$  AND  $\text{H}_2$  BIOWASTE  
SPECIFIC IMPULSE, 255 SEC ( $\text{CO}_2/\text{H}_2$ , 9/1)  
THRUSTER POWER, 112 WATTS EACH  
THRUST, 10 MILLIPOUNDS EACH  
POWER FOR 24 THRUSTERS AND FOR  
BIOWASTE COLLECTION, 6.7 KW

FIG. 10 - PROPOSED RESISTOJET SYSTEM FOR MANNED ORBITING RESEARCH LAB

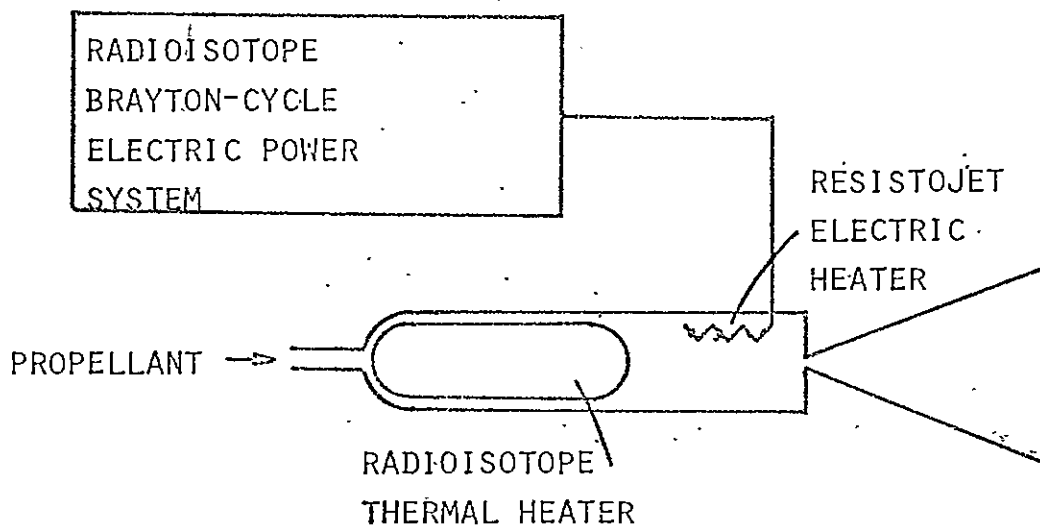


FIG. 11 - Hypothetical application of ATEP concept to MORL resistojet system.

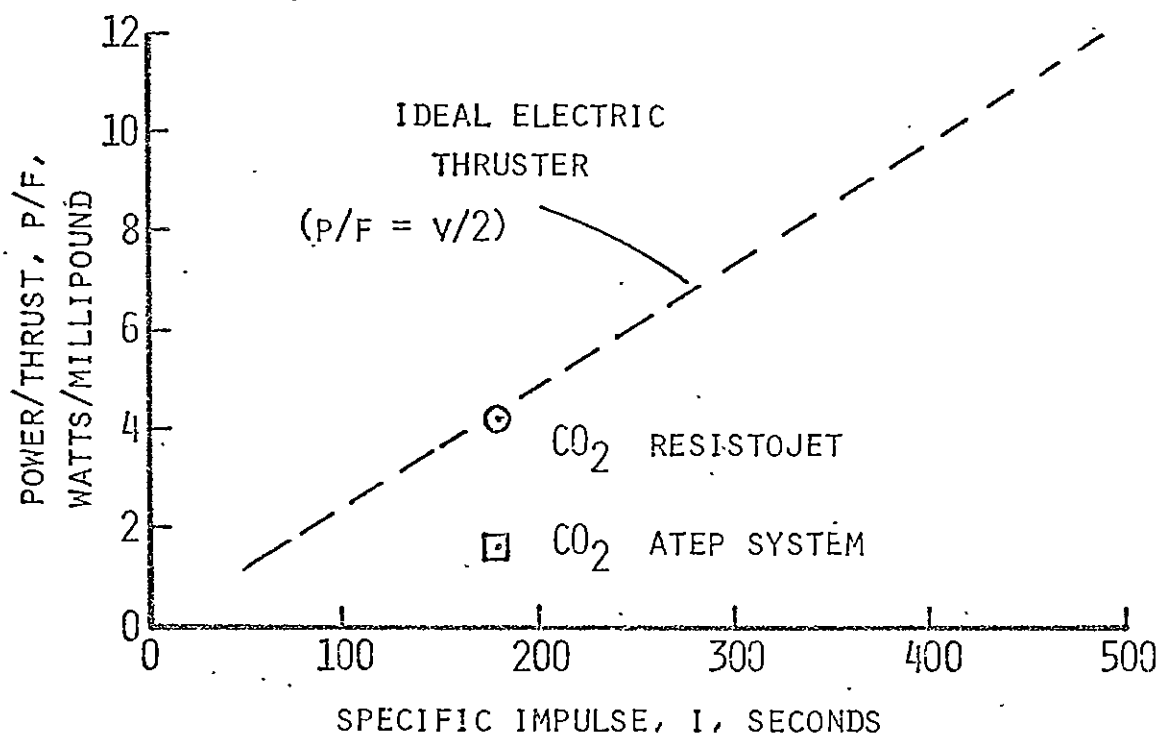


FIG. 12 - CO<sub>2</sub> resistojet performance with ATEP system.  
Radioisotope thermal-heating to 1200°K (2.46 thermal-watt/millipound) and electric heating to 3000°R (1670°K)



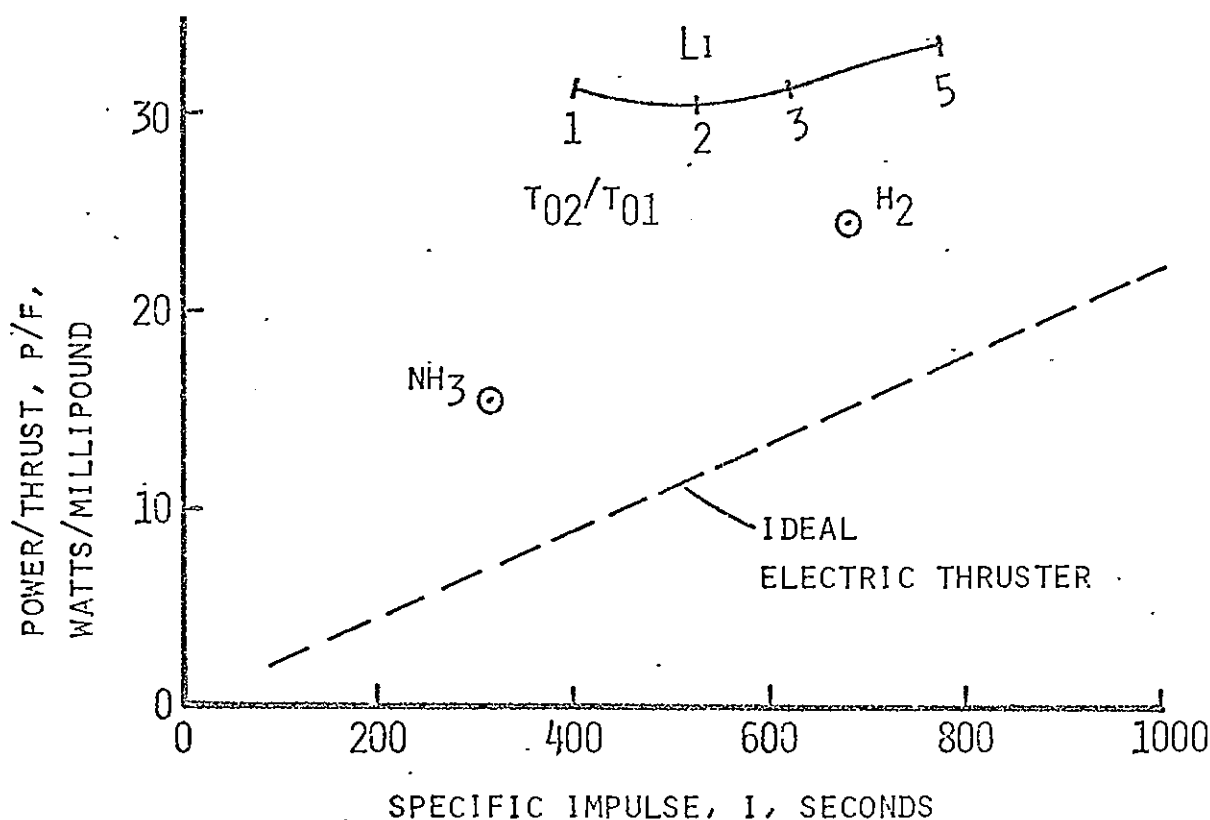


FIG. 13 - Theoretical performance of lithium resistojets with supersonic heat addition. Plenum temperature,  $T_{01} = 2500^\circ\text{K}$ ;  $A_2/A^* = 100$ . Ammonia and hydrogen data points for  $T_{01} = 2400^\circ\text{K}$ ,  $p_1 = 3$  atmospheres.

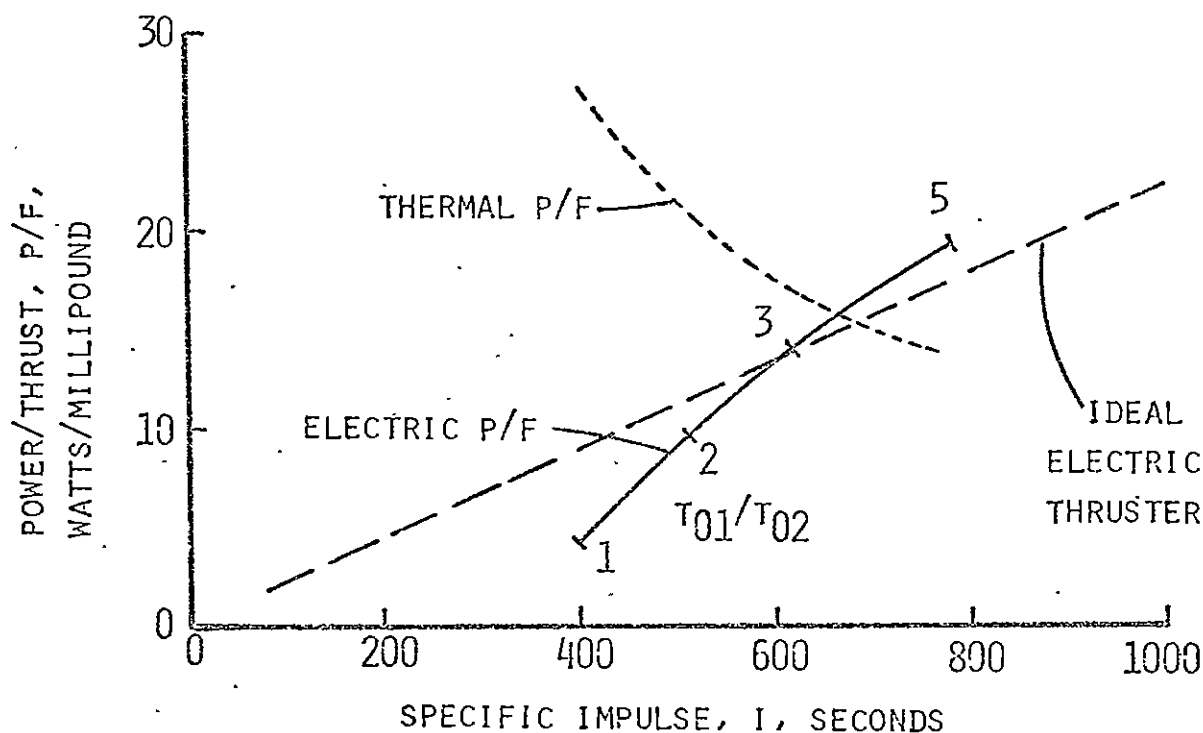


FIG. 14 - Theoretical performance of ATEP resistojets system with lithium propellant. Radioisotope heating to  $1200^\circ\text{K}$ , electric heating to  $2500^\circ\text{K}$  plenum temperature, electric heating of supersonic stream.

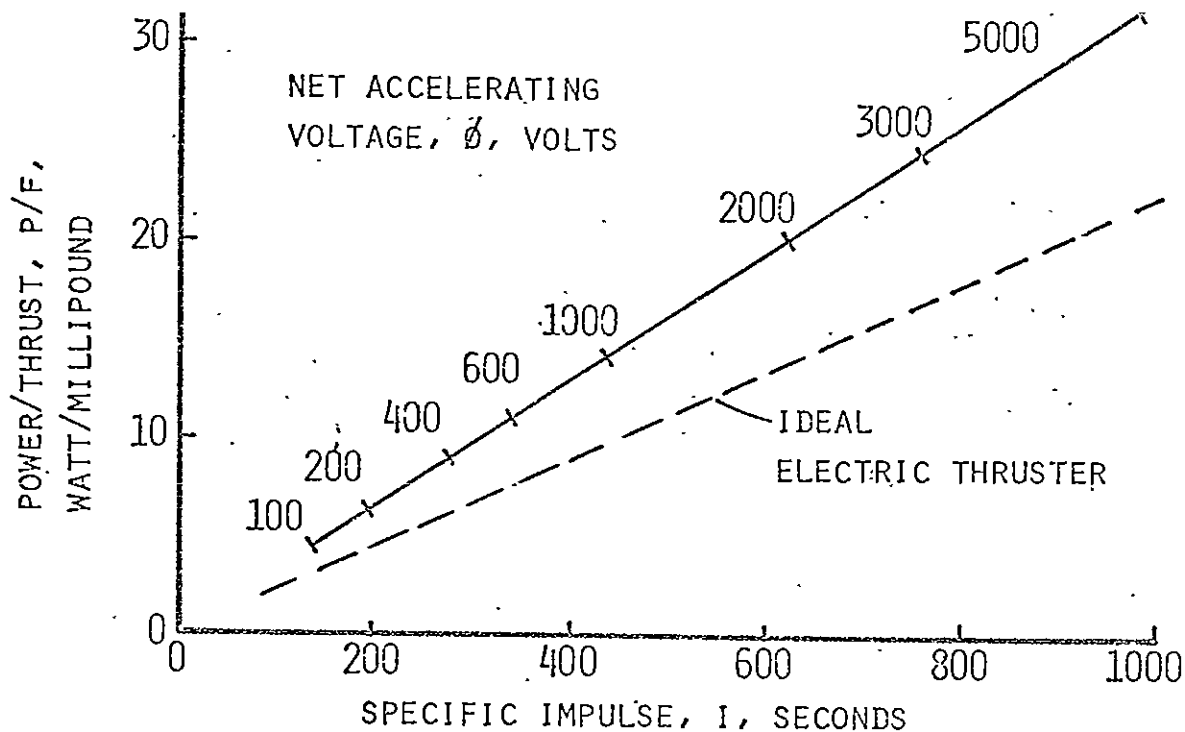


FIG. 15 - Performance of liquid-spray thruster for various net accelerating voltages. Charge/mass, 10,000 coulomb/kg neutralizer and feed-system power neglected (e.g., bi-pol design).

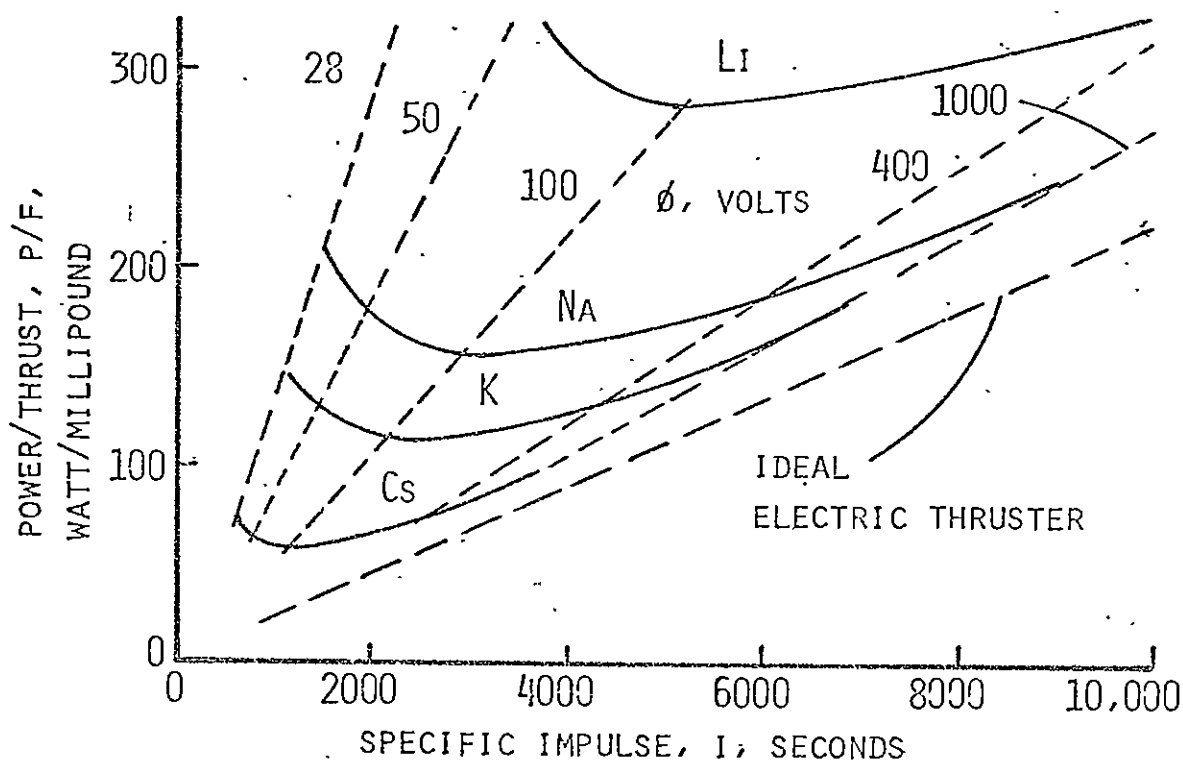


FIG. 16 - Performance of hypothetical ion thruster with various alkali propellants. Hollow cathode, composite grid, total throughput ion optics, 25 ev/ion neutralizer and feed power.

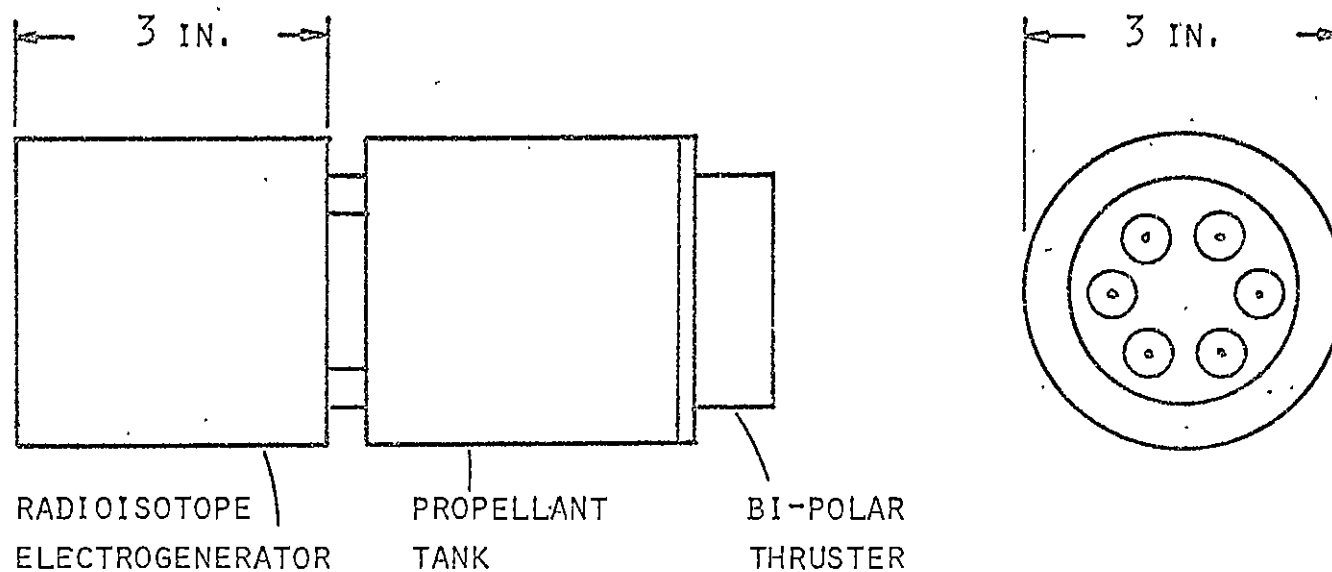


FIG. 17 - Hypothetical ACCENT system with liquid-spray bi-polar thruster. Thrust, 20-micropounds; specific impulse, 3000 sec.; total system weight, approx. 6 lb.; mission duration, 2-years; containment vessel, 1/8-inch tantalum; radioisotope, 27,600 curies of promethium-147.

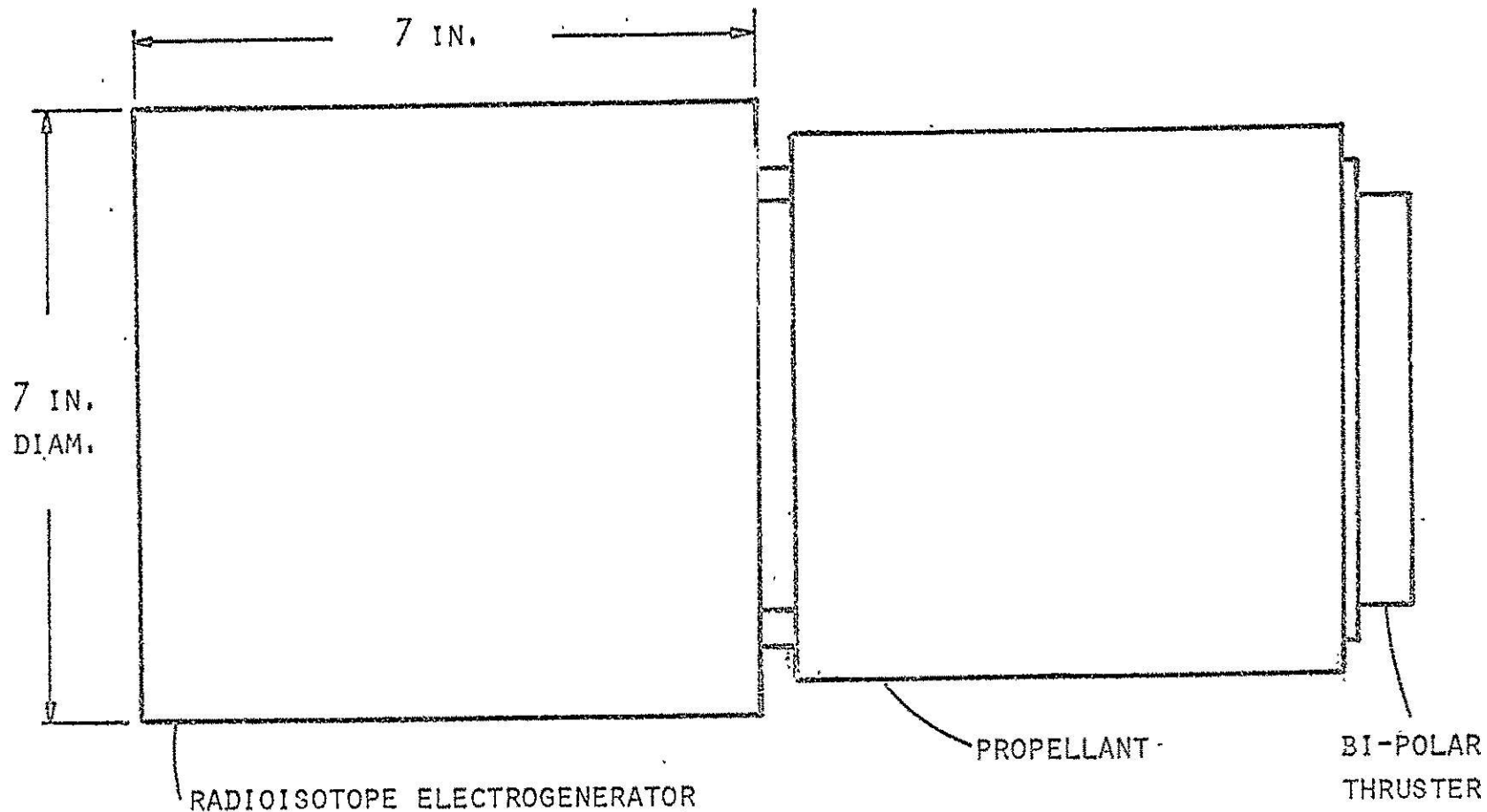


FIG. 18 - Hypothetical ACCENT system with liquid-spray bi-polar thruster. Thrust, 350-micropounds; specific impulse, 3000 sec.; total system weight, approx. 32 lb.; mission duration, 2-years; containment vessel, 1/8-inch tantalum; radioisotope, 483,000 curies of promethium-147.